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(OTV) ENGINE STUDY. PHASE A: EXTENSION
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ORBITAL TRANSFER VEHICLE (OTV)
ENGINE STUDY, PHASE A - EXTENSION
CONTRACT NO. NAS8-32996

BIMONTHLY PROGRESS REPORT NO. 3
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INTRODUCTION AND SUMMARY

The initial Phase A Orbital Transfer Vehicle (OTV) Engine Study Program was structured to identify candidate OTV engine cycle concepts and design configurations, to evaluate and assess the characteristics and capabilities of the candidates, and to determine an interim engine power cycle and engine configuration which can best meet the goals and requirements of the OTV program. In that initial portion of the study program parametric OTV engine data (performance, weight, cost) were generated and made available to OTV system contractors.

The OTV engine will be used to power the Orbital Transfer Vehicle that is carried into low earth orbit by the Space Shuttle. The OTV engine has the major objectives of high payload capability, high reliability, low operating cost, reusability, and operational flexibility. The OTV engine study is based upon 1980 technology. Preliminary cost data were also generated during initial Phase A studies.

Recognizing the reliability potential of the expander engine cycle and taking full advantage of continuing evaluation studies, through Phase B definition, by both vehicle and engine contractors, Rocketdyne recommended that both the staged combustion and expander engine cycles be continued through the OTV Vehicle Definition phase.

The current Phase A-Extension of the OTV engine study program will provide additional expander and staged combustion cycle data that will lead to design definition of the OTV engine. The proposed program effort will optimize the expander cycle engine concept (consistent with identified OTV engine requirements), investigate the feasibility of kitting the



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staged combustion cycle engine to provide extended low thrust operation, and conduct in-depth analysis of development risk, crew safety, and reliability for both cycles. Additional tasks will address costing of a 10K thrust expander cycle engine and support of OTV systems study contractors.

The detailed study objectives are to:

1. Prepare and submit a study plan for this extension
2. Perform pre-point design studies to optimize thrust chamber geometry and cooling, engine cycle variations and controls for an advanced expander cycle engine
3. Investigate the feasibility and design impact of kitting the staged combustion cycle engine to provide extended low thrust operation and identify the required new technology
4. Provide an in-depth analysis of development risk, crew safety, and reliability for both the staged combustion and advanced expander OTV engine candidates
5. Prepare a Work Breakdown Structure, Planning and Detailed Cost for a 10K advanced expander cycle engine
6. Provide engine parametric data book and support to the OTV systems studies contractors and define and clarify engine design characteristics and options
7. Prepare a comprehensive report at the conclusion of this study extension containing sketches, graphs, tables, technical details, and programmatic information resulting from the study.



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Previous contractual study efforts conducted by Rocketdyne and Aerojet (NAS8-32996 and NAS8-32999), and in-house sponsored programs conducted at Rocketdyne have provided a large data base for OTV-type engines in terms of both parametric and specific design point information. It is planned to make full use of these data to make the OTV study as comprehensive as possible.

The contracted extension effort includes five additional technical tasks and one reporting task. A scheduling for these tasks is shown in Fig. 1. The program began with an orientation briefing at NASA-MSC to discuss details of the work to be accomplished. At this briefing, Rocketdyne presented the approach of the program study plan, identified all tasks, their objectives, expected results, man-hour allotments, and program milestones.

As indicated in Fig. 1, the Task 8 report covering advanced expander cycle engine optimization was completed during this report period and effort continued on Tasks 9, 10, and 11. Task 12 contract support effort continued with release of a set of updated engine parametrics. Approximately 70 percent of the total planned man-hours have been expended in above efforts. This work is discussed in the main body of this report.

OTV PHASE A ENGINE STUDY EXTENSION

(NIAS 8-32996)

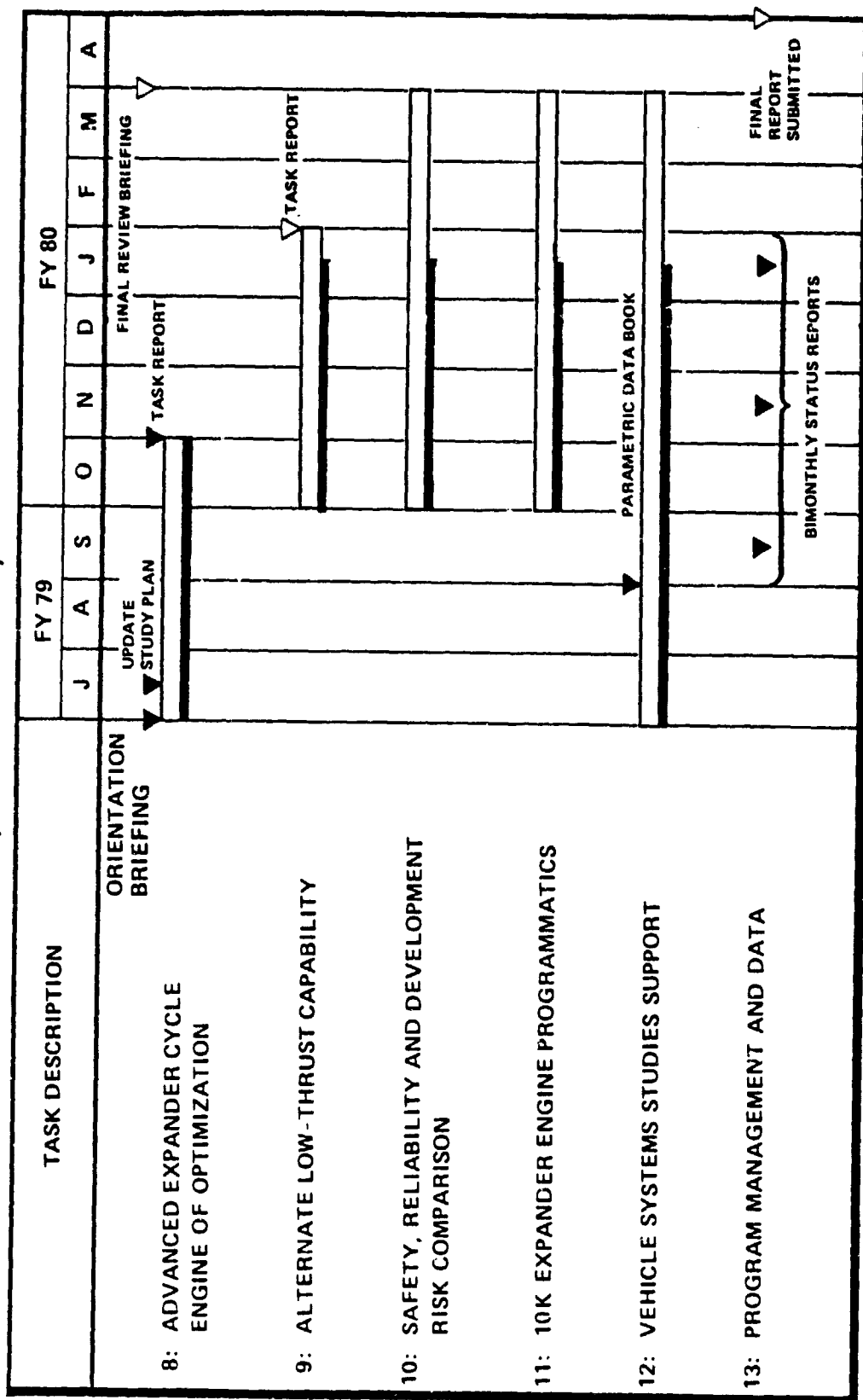


Figure 1. OTV Phase A - Extension, Program Schedule



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DISCUSSION

TASK 8. ADVANCED EXPANDER CYCLE ENGINE OPTIMIZATION

The overall objective of this task was to optimize the performance of the expander cycle engines with vacuum thrusts of 10K, 15K, and 20K with a maximum retracted length of 60 inches at a mixture ratio of 6:1. Maximization of payload delivery will be one of the primary goals of the study task for which performance/weight partials derived from NASA's TMX-73394 will be used during simplified mission analysis. The information generated will form the basis for subsequent point design studies.

A detailed report documenting the Advanced Expander Cycle Optimization effort has been released, R1/RD 79-346, December 1979. An addendum to R1/RD 79-346 which presents a more detailed discussion of 20K thrust level engines is currently being prepared. The results of Task 8 are discussed in detail for a typical 15K thrust level engine.

Selected Engine Configuration

The design characteristics of a typical 15K thrust engine are presented in Fig. 2. The engine selected uses 1980 state-of-the-art components to achieve a chamber pressure of 1610 psia. With this chamber pressure and within the extended engine length of 117 inches, the engine achieves an area ratio of 625 and delivers a specific impulse of 482.5 seconds. A flow schematic of the engine is shown in Fig. 3. Highlights of the component selections and component arrangements are indicated below.



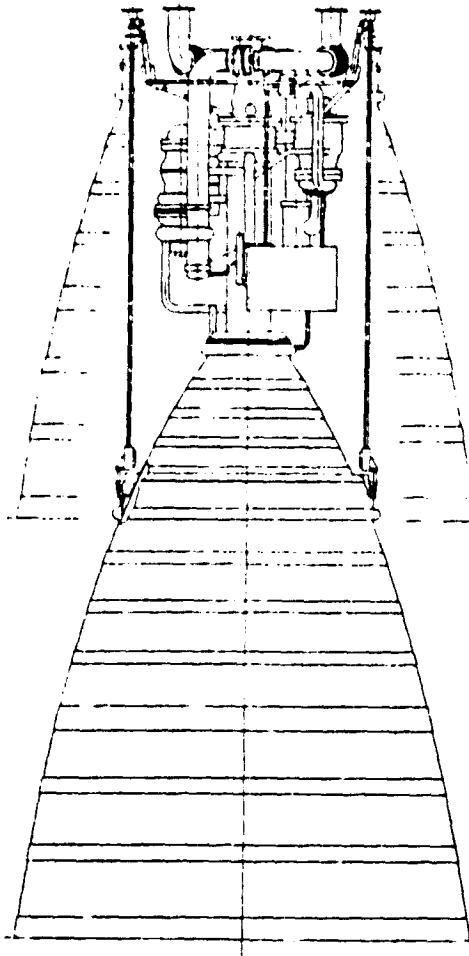
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BASELINE DESIGN
CHARACTERISTICS

● FULL THRUST (VAC), LB	15,000
● MIXTURE RATIO	6
● CHAMBER PRESSURE, PSIA	1610
● EXPANSION AREA RATIO	625
● SPECIFIC IMPULSE, SEC	482.5
● PUMP DISCHARGE PRESSURE, PSIA	
● OXYGEN	2682
● HYDROGEN	4621
● PUMP SPEEDS, RPM	
● OXYGEN	58,100
● HYDROGEN	107,140
● TURBINE INLET TEMP, R	955
● RETRACTED LENGTH, IN	60
● EXTENDED LENGTH, IN	117
● ENGINE DRY WEIGHT, LB	415
● TECHNOLOGY BASIS	1980

DESIGN FEATURES

- FULL-FLOW REGEN COOLING
- SERIES MAIN TURBINES
- GH₂-DRIVEN LOW PRESSURE PUMPS
- SMOOTH-WALL 20 IN. LONG COMBUSTION CHAMBER
- AUTOGENOUS TANK PRESSURIZATION

Figure 2. Performance Optimized Advanced Expander Cycle
Engine Baseline Design Characteristics

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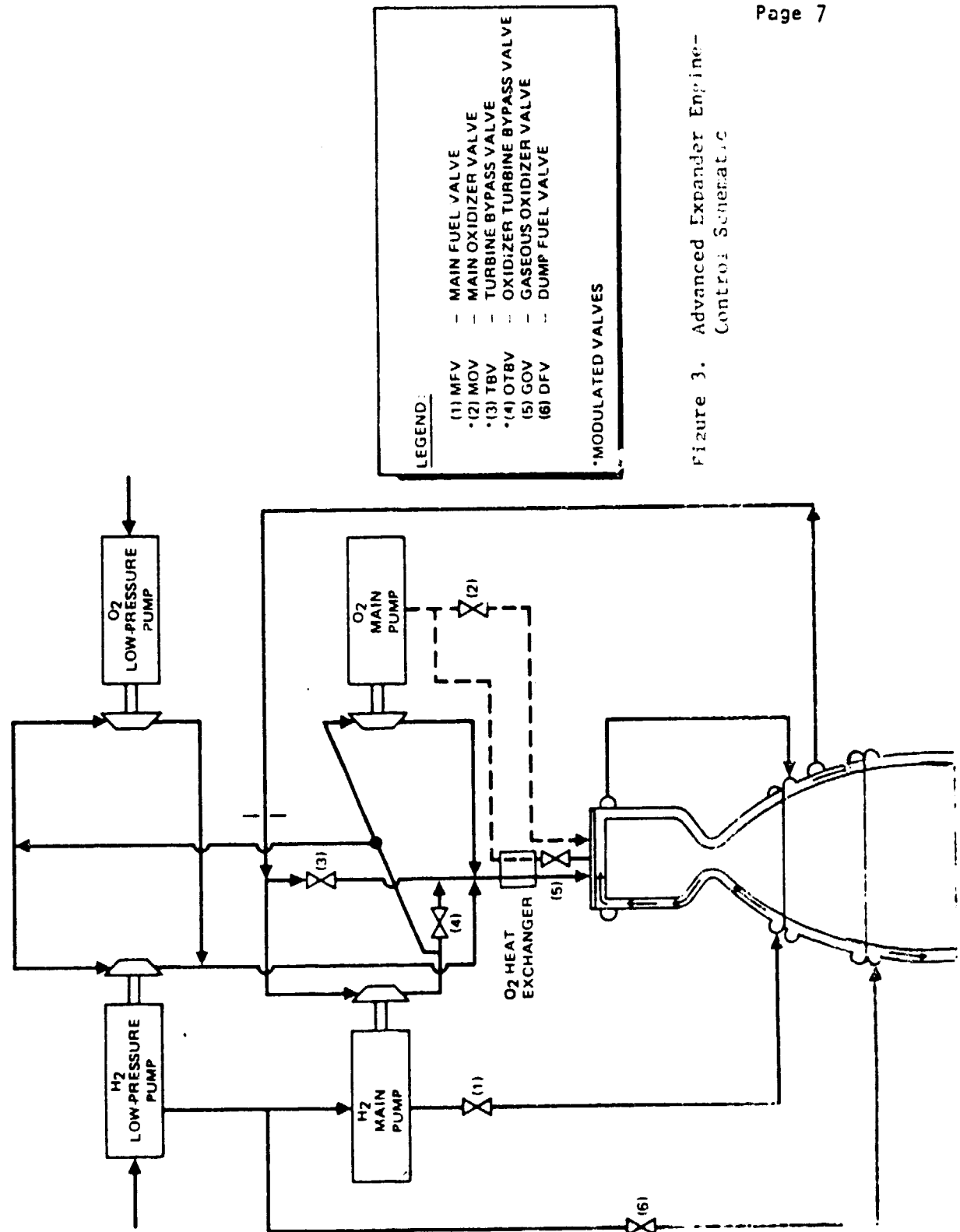


Figure 3. Advanced Expander Engine-
Control Schematic



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Power Cycle Configuration

Power cycle parameter optimization was carried out in the range of thrusts 10K-20K lbs. For each thrust, the cycle parameter optimization resulted in higher chamber pressures than earlier studies had shown, which increased specific impulse and decreased engine weight. The trends of increasing gravity losses with decreasing thrust forced the optimum payload to occur in each optimization case at or near 15,000 lbs. of thrust. Cycle optimization produced similar results at all of the thrusts investigated but was most pertinent at 15K lb thrust.

Four major areas of power cycle optimization were examined: main turbine arrangement, cycle energy source, high pressure pump design, and boost pump drive. Cycle configurations resulting from the first two areas were: parallel main turbines, series main turbines, turbine gas regeneration and turbine gas reheat. They were evaluated with smooth-wall combustors and with thermally enhanced combustor configurations.

Main Turbine Arrangement

The selected chamber pressure was obtained with optimized turbomachinery operating in a series main turbine arrangement (Fig. 3) for maximum utilization of available energy in the hydrogen coolant. A series turbine arrangement makes maximum use of the hydrogen flow available and, therefore, maximizes power in a flow limited power cycle like the topping expander cycle. Parallel main turbines were evaluated and found to yield chamber pressures 255 psia lower than the series main turbine arrangement.



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Cycle Energy Source

A smooth-wall combustor with a length of 20 inches from injector face to throat was selected as the main source of thermal energy input into the hydrogen coolant based on trade studies where turbine gas regeneration, turbine gas reheating, and combustion chamber thermal enhancement methods were evaluated as a means of augmenting the turbine gas inlet temperature.

High Pressure Pump Design

The fuel pump design is a three-stage centrifugal machine with inducer and axial inlet, similar to the successful design of the ASE. The fuel impellers are 3.5 inches in diameter with a tip speed and stage specific speed of 1630 ft/sec and 675 respectively. The oxidizer pump is a single stage centrifugal pump of 2.2 inches impeller diameter, 560 ft/sec tip speed and 1280 specific speed. Dimensions and speeds conform with 1980 state-of-the-art groundrules.

Boost Pump Drive

The impact of boost pump drive method upon overall engine performance is small because of the low horsepower requirements for these pumps, therefore other factors are more influential in the selection of the boost pump drive.

Preliminary selection of gaseous hydrogen turbine drive was made for both the hydrogen and oxygen boost pumps based on trade studies which considered performance, transient characteristics, flexibility of design, complexity, packaging and testing flexibility. These trade studies will continue in the Advanced Expander Point Design program to complete the evaluation and final selection. The other boost pump drives being considered are: electric, hydraulic and geared.



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Cooling Circuit. The cooling circuit selected is a full-flow regenerative scheme depicted in Fig. 3. All of the hydrogen flow, except for pump seal leakage and that used in supplemental cooling, is used in cooling the thrust chamber and nozzle in series arrangement. This hydrogen flow enters the combustor coolant jacket from the pump outlet at an expansion area ratio of 6:1 and it exits at the injector plane. From here it is routed externally and enters the nozzle coolant distributing manifold at an expansion area ratio of 6:1. The nozzle is then cooled in a one and one-half pass cooling scheme with the coolant exiting at an area ratio of 100. This circuit allows for the highest fuel turbine inlet temperature limits (1260 R) compatible with safe nozzle wall temperatures.

Injector. Coaxial injection elements and a transpiration cooled rigimesh faceplate have been selected for the baseline engine design. This configuration offers proven performance, reliability, and complete fabrication experience for minimum iteration during design and development.

Combustion Chamber. The combustion chamber is configured with a smooth hot wall and a length of 20 inches. The hot wall is regeneratively cooled with hydrogen flowing in axial passages from the throat region toward the injector face. The coolant channel geometry favors a contraction ratio of 4 which allows doubling the number of passages from the throat to the combustion chamber. The chamber configuration represents 1980 technology.

Combustion Chamber Alternate. One alternative to extended combustor length is a thermally enhanced thrust chamber which provides approximately 28 percent heat load enhancement in a short 17 in. long chamber. However, because heat transfer verification and low cycle fatigue (LCF) impact



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cannot be assessed with the same level of confidence as for the smooth wall design, the thermally enhanced thrust chamber concept is proposed as a possible alternate for future engine growth.

Fixed Nozzle. Configuration trade study results for the fixed portion of the nozzle reveal a highly developed state of current technology in this area. The selected design configuration has a thorough development basis in performance, reliability, fabrication and maintainability. It consists of a NARLOY Z channel cooled configuration in the high heat flux region near the throat plane and a thin wall A286 brazed lightweight tube structure from an expansion ratio of 6 to 210.

Nozzle Design. Because cooled metallic nozzles are considered current state-of-the-art, a hydrogen dump-cooled nozzle was selected for the initial baseline engine configuration. The retractable nozzle extends from an area ratio of 210 to 625. Cooling of this nozzle is feasible at all engine mixture ratios and only requires at off-design different coolant flow splits with other thrust chamber components than at on-design mixture ratio. No structural problems are expected with this type nozzle during start and shutdown transients, full thrust, or during normal handling.

Engine Control System. To enable control of thrust and mixture ratio level during mainstage and start, a simple three-valve closed-loop control system was defined. The valves required in this control system are shown schematically in Fig. 3. The control points selected have the capability to fully control the engine thrust and mixture ratio over the full range of required engine operating conditions. The control modulates the areas of the main oxidizer valve, the turbine bypass valve, and the oxidizer turbine bypass valve to achieve independently the proper balance of propellant flows for independent control of thrust and mixture ratio.



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The thrust and mixture ratio levels are variable upon vehicle command. Closed-loop control is required to maintain engine thrust and mixture ratio, within anticipated values of ± 5 percent and ± 2 percent, respectively of the commanded mainstage values.

Tank Head Idle

In tank-head idle the main propellant pumps are restrained from rotation while the low pressure pumps are allowed to operate and aid in the delivery of flow for thermal conditioning of the system. Initial trade studies indicate that it is desirable to have gas driven boost turbines which can start the boost pumps ahead of the main pumps to provide the latter with required NPSH and to aid in start conditioning.

Initial tradeoff also indicates that the heat exchanger required to vaporize the liquid oxygen prior to entry into the thrust chamber can be located in the turbine bypass circuit just upstream of the main injector.

Pump-Fed Idle

Pump-fed idle steady-state thrust is approximately 1800 pounds at a chamber pressure of approximately 200 psia. At this pressure, flow instability is not expected in the thrust chamber coolant jackets. Desired mixture ratios are maintained as high as cooling and power constraints allow (but not greater than 7:1) and will be established during Point-Design Studies. The engine will run in pump-fed idle mode until tank pressures providing NPSH of 2 and 15 are available for the oxidizer and fuel pumps respectively, at which time the engine can receive the mainstage thrust command.



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Mainstage start is initiated through pre-established ramping of the turbine bypass and main oxidizer valves. Ramp rates are determined to prevent excessive mixture ratio and thrust overshoots in the thrust chamber. Optimization of valve opening rates will be performed in the Point-Design Study with the transient model to achieve thrust chamber pressure and temperature environments compatible with engine prescribed cycle life. Engine shutdown transient requirements and valve operation will also be examined to obtain satisfactory results for maintaining engine life.

Engine Balance and Performance

Engine balance points at design mixture ratio of 6:1 and at design mixture ratio of 7:1 are presented in Table 1 for the selected concept.

Important features of the balance include:

- 10 percent fuel flow and 10 percent pressure drop reserves
- Full-flow regenerative cooling of both combustor and nozzle
- Dump-flow cooling of retractable nozzle
- Series main turbine arrangement
- Three-stage fuel pump and single-stage oxidizer pump
- Hydrogen gas driven low-pressure pumps

This design is the result of optimization to maximize engine specific impulse with resulting design values for chamber pressure, nozzle area ratio, and nozzle percent length for an engine retracted length of 60 in.



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TABLE 1 ENGINE DESIGN POINT BALANCE

Design Variables	MR = 6 Values	MR = 7 Values
<u>General</u>		
Thrust, lbf	15,000	15,000
Chamber Pressure, psia	1610	1568
Mixture Ratio, Engine	6:1	7.0:1
Mixture Ratio, Thrust Chamber	6.45:1	7.58:1
Specific Impulse (Engine), seconds	482.5	474.3
Specific Impulse (Thrust Chamber), seconds	483.4	475.0
Engine Fuel Flowrate, lb/sec	4.44	3.95
Thrust Chamber Fuel Flowrate, lb/sec	4.12	3.64
Engine Oxidizer Flowrate, lb/sec	26.62	27.67
Thrust Chamber Oxidizer Flowrate, lb/sec	26.58	27.63
Dump Coolant Fuel Flowrate, lb/sec	0.28	0.28



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TASK 9. ALTERNATE LOW THRUST CAPABILITY

The feasibility and design impact on the 20K staged combustor of the requirement that the engine be adaptable to extended low-thrust operation (1-2K) has been studied during this report period. This task has been addressed from the following aspects:

1. Attainable specific impulse at the low thrust level
2. Thrust chamber cooling capability at low thrust
3. Injector modification or kitting
4. Combustion stability consideration
5. System transient analysis and control requirements
6. Feed system coupled low frequency stability analysis
7. Possible turbopump modifications to alleviate feed system coupling
8. Dump cooled nozzle operation
9. Possible turbine-nozzle flow area modification

Kitting of the basic engine to permit extended low-thrust operation has been examined and the following options have been considered:

1. Removal of the preburner and the lox feedline and valve to the preburner to reduce the hydrogen side flow resistance during low thrust operation
2. In the main injector increasing the lox side pressure drop by kitting a new injector or by using inserts into the present injector elements



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3. Modification of the fuel pump impeller to use more highly backswept blades to improve the head-flow characteristic. This change would avoid a possible feed system coupled instability at low thrust
4. Possible use of fuel pump recirculation in order to avoid the positive slope region of the fuel pump H-Q curve
5. The use of a turbine bypass valve for control purposes

The results of the studies indicate that extended low thrust operation of the 20K staged combustion cycle engine is feasible with the current configuration at mixture ratio up to 2.1. With modification (kitting) of the main injector and modification of the fuel pump impeller the engine can operate at mixture ratios up to 6.0. The detailed results of this task will be presented in a Task Study Report to be issued in February 1980.

TASK 10. SAFETY, RELIABILITY AND DEVELOPMENT RISK COMPARISON

The objective of this task is to perform comparative analyses in the following areas: (1) crew safety, (2) mission success, and (3) development risk with respect to DDT&E program schedule advances or slippages.

During the report period the areas of crew safety and mission success were investigated in depth to provide a perspective for development risk comparisons of candidate OTV propulsion systems, and as an input for programmatic analyses. Historical data were assembled to show the reliability and safety requirements for engines used in manned vehicles of past space programs. The requirements were compared to those of the current Space Shuttle program. The following conclusions were drawn:



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1. Engines for Past Manned Vehicles

Safety considerations for engines of the Mercury, Gemini, and Saturn/Apollo programs relied mainly on high demonstrated engine reliability goals. Safety was addressed on a vehicle and program level but not on an engine level. The Mercury and Gemini engine programs required an 85% reliability goal to be demonstrated at a 90% confidence level. The Saturn/Apollo program required 99% reliability to be demonstrated at 50% confidence for the H-1, F-1, and J-2 boosters. Fig. 4 shows the demonstrated reliabilities for the J-2 and F-1 engines as a function of total number of tests and calendar time. A reliability exceeding 99% was allocated, but no demonstration required, for the SPS and LEM engines. No propulsion system redundancy existed except for the H-1 boosters, and for the J-2 in the 2nd Saturn stage.

2. Engines for the Space Shuttle

Safety is addressed as an important issue for all aspects of the Space Shuttle engine programs. Engine reliability is not required to be numerically demonstrated. High reliability is achieved through a combination of measures such as the Design Verification Specification, demonstration of engine service life, fail safe design criteria, and specific design requirements. Propulsion system redundancy is considered as a major part for safe return of the crew.

Several NASA documents originating at MSFC, JSC and HQ were reviewed for information regarding engine requirements for manned space flight (see

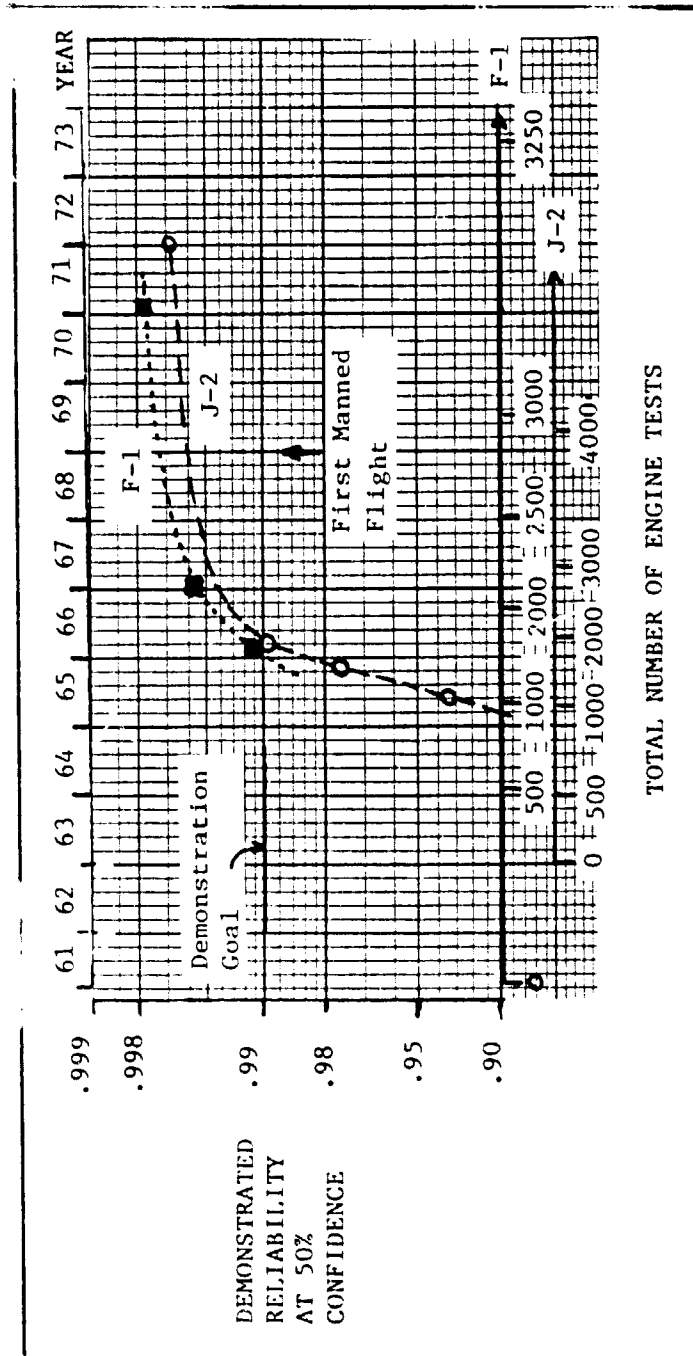


Fig. 4 Demonstration F-1 and J-2 Reliabilities



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Table 2). Though the documents revealed useful, general requirements, no specific requirements applicable to OTV engines during space flight were found.

OTV vehicle propulsion system configurations were investigated with respect to probability of mission success and of safe return. The parametric analysis considered the following influences: main engine redundancy, availability of a low thrust Auxiliary Propulsion System for backup, single engine reliability, main engine component redundancy, and main engine cycle.

Specific engine design approaches recommended for high OTV propulsion system reliability to enhance mission success and crew safety were established.

The study results of this task are presently being assembled in the form of a presentation for the Orbital Transfer System Studies Coordination Meeting in Huntsville, Alabama on January 24, 1980.

TASK 11. COST AND PLANNING COMPARISON

A WBS was developed for the 10K thrust expander engine. Costs for each WBS second level item were estimated, based on a modification of the original 20K thrust staged combustion cycle cost evaluation. Work was started on a DDT&E schedule for the 10K thrust expander. The relationship between engine reliability and engine cost, determined as part of Task 10, will have a significant effect on the results of Task 11.



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TABLE 2. NASA DOCUMENTS REVIEWED FOR SAFETY AND RELIABILITY

Document No.	Title	Year of Issue	Main Safety & Reliability Requirements
M-D MY 1700.120 (MSFC)	Safety Program Directive No. 1	1969	<ul style="list-style-type: none">• Hazard Category Definition and Hazard Analysis Instructions• Preparation of System Safety Plan• Defense In-Depth Through Hazard Reduction Precedence Sequence (Preferred Order of Design Measures)
85M03885 (MSFC)	Guidelines for Performing FMEAs on Space Shuttle	1971	<ul style="list-style-type: none">• Instructions for FMEAs• Criticality Category Definition and Criticality Analysis Instructions
NHB 5300.4	Safety, Reliability, Maintainability & Quality Provisions for Space Shuttle	1974	<ul style="list-style-type: none">• Similar to M-D MY 1700.120
JSC 11123	STS Payload Safety Guidelines Handbook	1976	<ul style="list-style-type: none">• Detailed Design Guidelines for Safety
JSC 0700, Vol. X	Space Shuttle Program: Space Shuttle Flight & Ground System Spec.	1978	<ul style="list-style-type: none">• Shuttle Subsystem Redundancy No Less Than Fail Safe During Entire Mission• Operational Status Verification of Redundant Subsystems• NHB 5300.4 Applies
NHB 1700.7	Safety Policy & Requirements for Payloads Using the STS	1979	<ul style="list-style-type: none">• Baseline Policy for STS Payloads• Payload Single-Failure Tolerant for Critical Hazards• Payload Double-Failure Tolerant for Catastrophic Hazards• Three Propellant Valves in Series
JSC 13830	Implementation Procedure for STS Payloads System Requirement	1979	<ul style="list-style-type: none">• No Safety Criteria, Only Procedures and Instructions